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Technical Memorandum No. 33-43

Electric Spacecraft For Planetary and Interplanetary Missions

L. D. Jaffe
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JET PROPULSION LABORATORY
CALIFORNIA INSTITUTE OF TECHNOLOGY
PASADENA, CALIFORNIA

March 15, 1961

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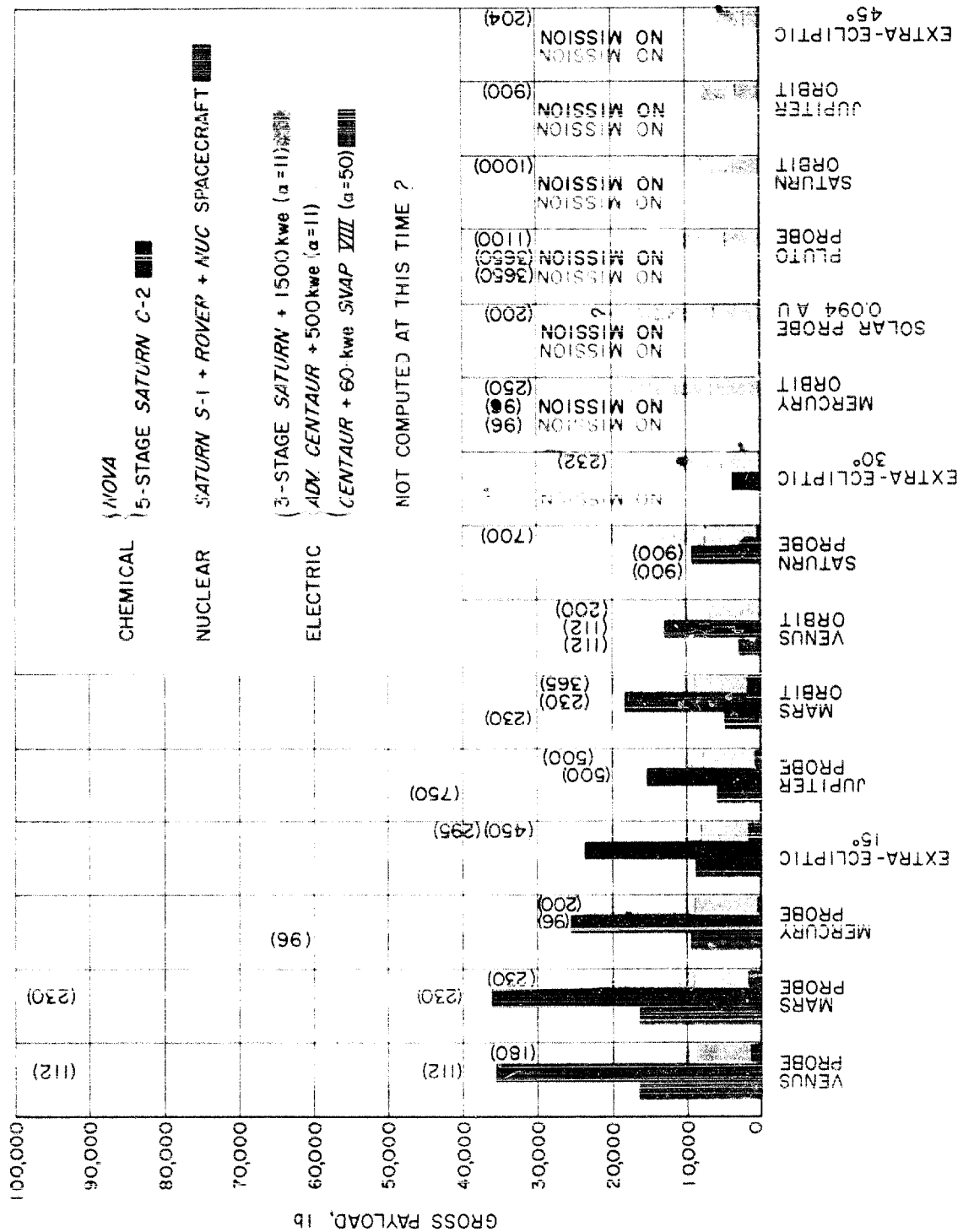
JPL Technical Memorandum No. 33-43, Revised Figure 1.
Performance Comparison: Chemical vs Nuclear vs Electric



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January 24, 1962

Recipients of Jet Propulsion Laboratory
Technical Memorandum No. 33-43

GROUP 4
Downgraded at 3 year intervals
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SUBJECT: Errata for TM 33-43

Gentlemen:

It is requested that the enclosed curve be substituted for Figure 1 of JPL TM 33-43. The following comments should be included:

The gross payload capability of the Saturn S-1 + Rover + nuclear spacecraft (shown in Figure 1 of TM 33-43) for various interplanetary missions is low because of an unnecessary limitation on the initial acceleration of the second nuclear stage. An acceleration of approximately 10^{-2} g was utilized, whereas the optimum is approximately 0.3 - 0.5 g. By employing the higher initial acceleration, the gross payloads are increased as indicated by the enclosed figure. These results supplant those of the initial Report.

It should be noted that, although the gross payload for the less energetic planetary and interplanetary missions is increased, there is no increase in the number of missions which can be performed using the optimum acceleration. Thus, the conclusion regarding the application of nuclear systems for unmanned interplanetary and planetary missions is not altered; i.e., advanced electric systems are preferable to the nuclear systems.

Very truly yours,

N. F. White, Assistant Manager
Reports Section

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ABSTRACT

In a preliminary comparison of the performance capability of chemical-, direct (Rover-type) nuclear-, and electric-powered spacecraft for fifteen planetary–interplanetary missions of interest to space scientists, advanced electric spacecraft were found to have a unique advantage for a large fraction of these missions; this performance advantage appears more favorable if the spacecraft is powered by thermionic rather than turbogenerator reactor systems. In addition, it is shown that thermionic power conversion would have distinct advantages over turbogenerator power-conversion systems with respect to utilization and compatibility with other operating components in the spacecraft. Because problems exist in developing a flyable thermionic reactor, primarily in the materials and lifetime areas, a final choice between the thermionic and turbogenerator conversion cannot be made today. However, the above incentives and the state of the thermionic art are such as to warrant a highly accelerated research and development effort in order to permit (1) an early choice between thermionic and turbogenerator conversion systems and (2) utilization of the preferred type in planetary spacecraft during this decade.

I. INTRODUCTION AND PERFORMANCE ESTIMATES

For nearly a year, the Jet Propulsion Laboratory has been investigating the feasibility of utilizing electric propulsion systems in spacecraft for deep-space missions. Early studies were oriented toward an understanding of their unique characteristics and spacecraft integration problems as well as a preliminary evaluation of the *Snap VIII* propulsion system for a spacecraft with a Mars orbiter mission or a probe out of the plane of the ecliptic. In October 1960, an internal interim report (Ref. 1) was issued to summarize the status of the studies.

It quickly became apparent that the *Centaur–Double Snap VIII* vehicle will be limited in its capability to perform scientific missions in deep space. Consequently, more recent studies have focused on the characteristics and time schedule for more advanced electric systems and those which might supplant the *Centaur–Double*

Snap VIII for most of the planetary and interplanetary missions.

The approach to the problem has consisted of the following:

1. Determine the space-science missions that are desirable for the next two decades. Include, if possible, today's estimate of the relative desirability and magnitude of those missions; i.e., the fraction of available vehicles that would be devoted to each type of mission.
2. Estimate the net or scientific payload capability of existing and planned vehicles for those missions.
3. Determine the relative advantages of appropriate vehicles and their state of development, and estimate the time schedule for availability.
4. Narrow the field of vehicle systems and recommend one course, or possibly alternate courses, of action.

The present Report covers a less ambitious area than that indicated, but studies have progressed to the point that preliminary information can be provided and pertinent conclusions can be drawn.

Before discussing performance capabilities, it seems desirable to define several terms and indicate the assumptions made. Gross payload will refer to the weight of the spacecraft at its destination less the weight of the propulsion power plant:

$$W_{PA} = W_{s/c} - W_{PP}$$

Thus, the gross payload normally includes the weight of the scientific instruments, telecommunications, guidance and control, and their associated structures. This is true for the chemical spacecraft; the gross payload of the electrical spacecraft, however, does not include power for telecommunications—an arbitrary penalty in the comparison. For some missions, such as planetary orbiters or Pluto probes, this telecommunications power requirement could be as large as 1 to 5 kwe and usually for long operating periods. At a typical weight of 300 lb/kwe, the chemical system would thus require 300 to 1500 lb more gross payload in spacecraft than would the electrical system for an identical scientific payload.

For the electric systems it is convenient to refer to the power-plant weight in terms of its electrical gross power level and specific weight α; thus,

$$W_{PP} = \alpha P_G$$

The specific weight α usually ranges from about 5 to about 100 lb/kwe for propulsion systems. Discussions will exclude secondary power and will be confined to an electric propulsion system that consist basically of the following: (1) a nuclear reactor which feeds, (2) a power conversion system, either dynamic or static, which converts the thermal energy of the reactor to electric energy, and (3) an electric thrust unit with its associated propellant tank and controls. The performance calculations are based on ion thrust units, but subsequent discussions on the power-conversion systems are applicable to magnetohydrodynamic and other electric thrust units as well.

Performance calculations for the chemical and nuclear vehicles were made by N. Haynes, of this Laboratory, with some design criteria for the nuclear vehicle by D. Spencer. Spencer assumed a Rover-type low-thrust system with an I_{sp} of 900 sec and an empty weight that was 0.2 of the spacecraft gross weight; Rom and Johnson of the Lewis Research Center discussed a similar low-thrust system in a paper presented at Los Alamos in October, 1959 (Ref. 2).

In the chemical vehicles, Haynes also considered a hypothetical five-stage *Saturn*, because the four-stage *Saturn* would be incapable of performing many of the missions; the extra stage should increase capability for high-energy tasks. Because a complete redesign of the *Saturn* for only a limited number of high-energy missions seemed impractical, it was assumed that the payload of the four-stage vehicle was divided between a fifth chemical stage and some new, smaller payload; in accordance with classical theory, weights were distributed to give equal incremental velocities in the fourth and fifth stages.

The *Nova* was conceived as a first stage, with six F-1 engines and liquid hydrogen–liquid oxygen in the second and third stages. Such a vehicle would have a launch weight of approximately 7 million pounds and a payload capability of about 300,000 lb in a 300-nm Earth orbit. A fourth stage with LOX–H_2 was used to achieve Earth escape. In the case of the orbiter missions, Haynes used solid-propellant retrorockets with a specific impulse of 300 and propellant to gross-propulsion weight ratio of 0.9.

The electric missions were computed by W. Melbourne (Ref. 3) of this Laboratory, using constant thrust directed tangentially in the spiral trajectories and an Irving and Blum optimum-thrust program in the heliocentric trajectory. Consequently, payload weights are slightly optimistic for future practical missions—probably 5 to 10%. Table 1 indicates the specific weight assumed for the electric propulsion systems; additional details are given in Part II, which compares thermionic and turbogenerator systems.

Figure 1 summarizes gross-payload estimates for some chemical, a direct (Rover-type) nuclear, and some electric spacecraft for planetary and interplanetary missions. Earth satellites and lunar missions have not been considered. The *Advanced Centaur*, a vehicle in the proposal stage only, was included in order to see what planetary performance an electric spacecraft for a vehicle with a 15,000-lb Earth orbit capability would have.

Gross payload ranges from 0 to 100,000 lb for 15 interplanetary missions for which space science has evidenced marked interest. They consist of seven probes or flybys (to the Sun, Mercury, Venus, Mars, Jupiter, Saturn, and Pluto), five planetary circular orbiters for Mercury, Venus, Mars, Jupiter, and Saturn, and extra-ecliptic probes with trajectories inclined at 15, 30, and 45 deg to the plane of the ecliptic. M. Neugebauer (Ref. 1) has pointed out the unusual scientific interest and value in experiments conducted well outside the plane of the ecliptic. It should be noted that, in the case of solar probes, scientific interest rises very rapidly as the spacecraft becomes capable of reaching or penetrating the inner solar corona (20 solar radii or 0.094 AU).

One other class of missions especially interesting to space scientists, the planetary orbiter–lander, has not been computed as yet. However, the orbiter–lander will be a logical extension of orbiters; presently shown orbiter payloads should give some indication as to whether sufficient weight is available in them to permit construction of a capsule that would be ejected from the orbiting spacecraft for a landing. Note, therefore, that the ability to perform orbiter missions may be doubly valuable if it appears that the spacecraft would be capable also of the orbiter–lander missions at a later date.

Table 1. Weight breakdown of propulsion system for *Centaur* 300-kwe spacecraft

Component	Thermionic system lb/kwe	Turbogenerator system lb/kwe
Reactor conversion system (described in Part II)	10.8	15.2
Thrust unit, propellant tank, associated plumbing, and controls	1.5	1.5
Contingency	1.0	1.3
Total	13.3	18.0

NOTE: The 500- and 1500-kwe propulsion systems for the *Advanced Centaur* and *Saturn* were assumed to have thermionic conversion. A specific weight of 11 lb/kwe was estimated for the 500-kwe propulsion system; three of these units would be clustered for the 1500-kwe *Saturn* to avoid development of a separate 1500-kwe reactor conversion unit.

Missions have been listed in Fig. 1 approximately in the order of increasing difficulty. Vehicles studied are shown in the legend:

1. The chemical systems, in green, include the *Nova* and a five-stage *Saturn C-2*, with estimated capabilities of 300,000 and 45,000 lb in a 300-nm Earth orbit, respectively.

2. The *Saturn* nuclear vehicle, in red, consists of a Rover-type nuclear stage on top of the *S-1 Saturn* first stage and is estimated to have a payload capability of about 79,000 lb in Earth orbit; the spacecraft would have a second Rover nuclear power plant, but of lower thrust, for the deep-space missions shown.

3. The electric vehicles are coded brown and include (a) a three-stage *Saturn C-2* booster with a 1500-kwe electric spacecraft, (b) an *Advanced Centaur* with a 500-kwe electric spacecraft, and (c) a standard *Centaur* with a 60-kwe *Double Snap VIII* electric spacecraft. The *Centaur* was assumed to be capable of placing 8800 lb* in 300-nm orbit.

The designation "No Mission" signifies either that the propulsion system is incapable of delivering any weight to its destination or that the gross weight was less than that required for spacecraft guidance, control, telecommunications, structure, and a minimum of scientific instruments. For example, the *Nova* can perform "no useful scientific mission" in a Mercury circular orbiter, because its gross payload of approximately 500 lb would not even provide weight for guidance, telecommunications, and structure.

The number at the top of each bar graph indicates total transit time from launch to final mission destination. The graphs for chemical (or electric) vehicles are superimposed; i.e., are all referenced to zero. In the case of the orbiter missions, performance was calculated for circular orbits at an altitude of 500 mi for Mercury, Venus, and Mars and 2000 mi for Jupiter and Saturn.

Payloads for the chemical *Nova* are unusually high for the low-energy missions (about 50,000 to 100,000 lb) but rapidly drop off and become impractical or nonexistent for a Mercury circular orbiter, a Pluto probe, Saturn or Jupiter circular orbiters, a solar probe at 20 radii, and the 30- and 45-deg extra-ecliptic missions. The Mercury orbiter mission, as indicated, turns out to be much more

*Since these calculations were started, NASA has indicated that the *Centaur* can place about 8500 lb of payload in 300-nm orbit.

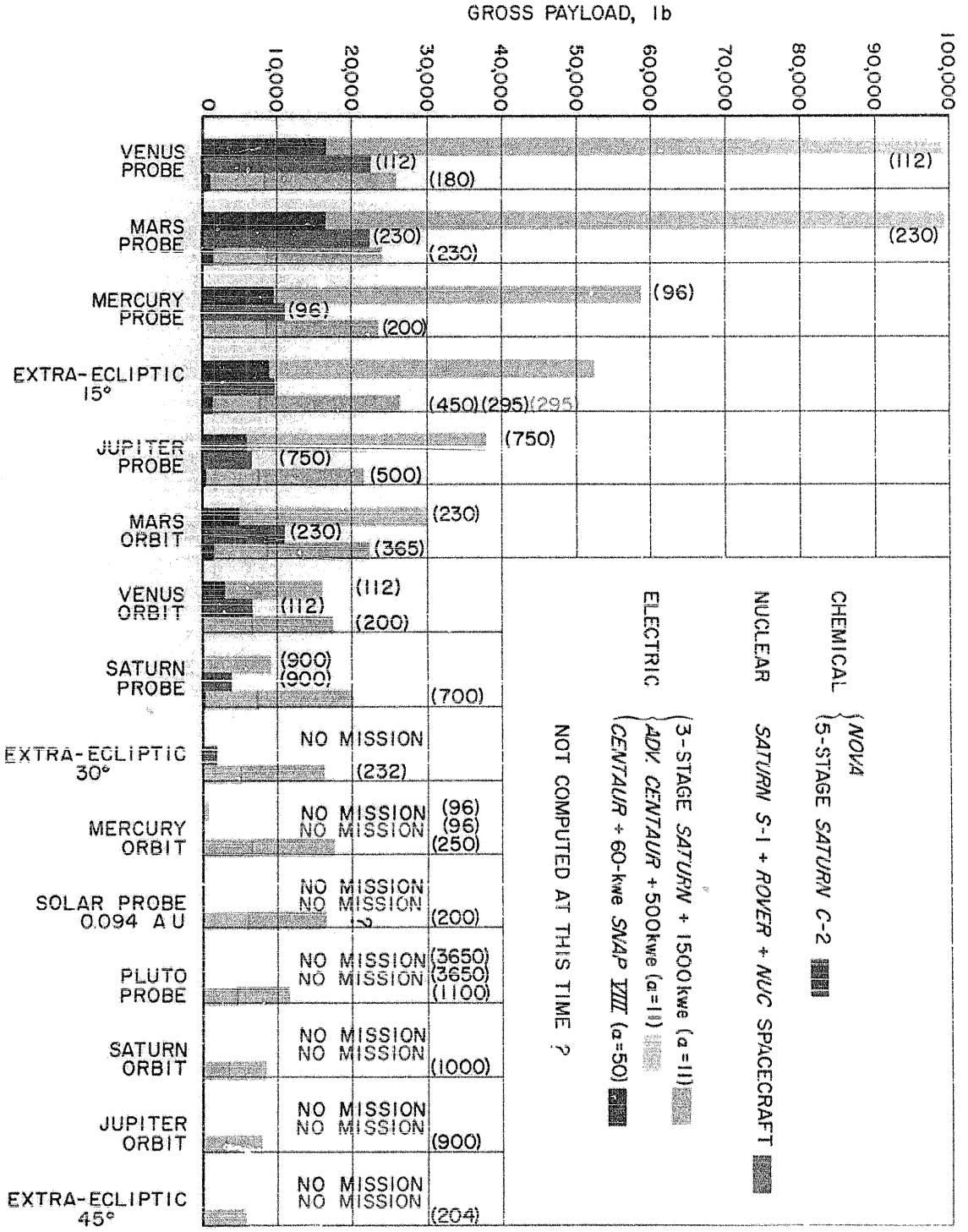

Fig. 1. Performance comparison: chemical vs nuclear vs electric

difficult to perform than might be expected because of the extremely large excess spacecraft velocity that must be removed at encounter in order to go into orbit; the orbital velocity of Mercury itself, of course, is very high (29.75 mi/sec mean). If Mercury and Jupiter orbiters with large eccentricities and low perigees should prove to be acceptable to space scientists, *Nova* might provide meaningful missions in these cases also.

The five-stage *Saturn* tends to parallel *Nova* at a smaller capability. In this case, the probe to the planet Saturn also produces no payload for the indicated transit time; however, longer times may permit missions with small payloads.

Although the three-stage *Saturn C-1* and *C-2* vehicles are not shown, it is of interest to note that the performance of the former would confine its use to Venus and Mars flybys (3000- and 3200-lb payloads, respectively) and the latter to flybys and orbiters for Venus and Mars. Payloads for the three-stage *C-2* would be about 60 to 65% of those for the five-stage *C-2* for the respective four missions.

The nuclear systems of the Rover type have very little to offer for planetary and interplanetary mission work. The *Nova* has greater payload capability for low- and medium-energy missions, whereas the electric systems are superior for high-energy missions.

The brown electric graphs indicate that the *Saturn* electric spacecraft can probably perform all of the indicated missions, even those that *Nova* cannot. In fact, the *Advanced Centaur* electric system can probably perform most, if not all, missions, provided a power-plant specific weight of 11 to 13 lb/kwe can be developed. Note, too, that the payloads for the *Advanced Centaur* are large enough so that, at least for the nearer planets, where they are needed most, meaningful orbiter–lander missions can probably be performed.

The *Centaur–Double Snap VIII* is limited in electrical power and can perform, at most, three or four of the missions in the designated transit time if the specific weight is 50 lb/kwe. Longer times, approximately twice those shown, seem to result in a greater number of feasible missions; but further study is needed for verification. This does not imply that the present development emphasis on *Snap VIII* should be changed. Regardless of its performance limitation, *Snap VIII* is well along in its development and is very important in advancing electric-propulsion technology. In addition, it serves as a going project and milestone to encourage development work on a useful time schedule.

Figure 1 indicates the broad picture; but it seems desirable to look closer at vehicles that could be directly applicable to a planetary program late in this decade. One might ask whether any standard *Centaur*-based electric spacecraft is capable of performing most of the foreseeable planetary missions or whether it is necessary to use the larger, costlier *Saturn* as booster.

Figure 2 compares the four-stage chemical *Saturn C-2* with a *Centaur* 300-kwe electric vehicle. The Earth orbit capability of the former was assumed to be about 45,000 lb, whereas that for the *Centaur* booster would be only about 8800 lb.

Gross payload now ranges from 0 to 10,000 lb for the same 15 missions. Specific weights of 18 and 13 lb/kwe represent estimates at this time of what can be achieved with a turbo-alternator dynamic conversion system and the thermionic static system, respectively, at a 300-kwe power level (see Table 1). By reducing the specific weight α of the power plant from 18 to 13 lb/kwe in a spacecraft with a 300-kwe power plant, the weight saved (1500 lb) can be put directly into the *scientific payload*. If one estimates that guidance and control, telecommunications, and spacecraft structure will account for about 1000 to 1500 lb, then the balance of the gross payload—the scientific payload—would often be doubled or even quadrupled. Alternately, this reduction in specific weight could result in enough increase in payload to permit an orbiter–lander mission.

Figure 2 shows that the *Saturn* chemical vehicle, with a 12,000- to 13,000-lb payload, is markedly superior to the *Centaur* 300-kwe vehicle for Venus and Mars probes. The *Centaur*-electric, with a specific weight of 13 lb/kwe, would surpass it in all other missions except the Mercury probe, however. Note that the specific weight must be down to about 11 to 13 lb/kwe in order that most of these missions may be performed successfully with the indicated vehicles. Thus, the system at 18 lb/kwe could not perform a Pluto probe, a Saturn or Jupiter orbiter, or the 45-deg extra-ecliptic probe; and it probably could not perform a 30-deg extra-ecliptic probe mission.

Table 2 summarizes the mission capability for the same three vehicles. The mission was not considered feasible if the gross payload was less than 1000 lb and feasible if it was greater than 1500 lb. The *Saturn* cannot perform eight of 15 missions and would probably be marginal in one other; six of the 15 seem feasible. However, two of these six are Mars and Venus probes; because these missions will have been performed by the time the four-stage *C-2* becomes available, they are of limited interest only.

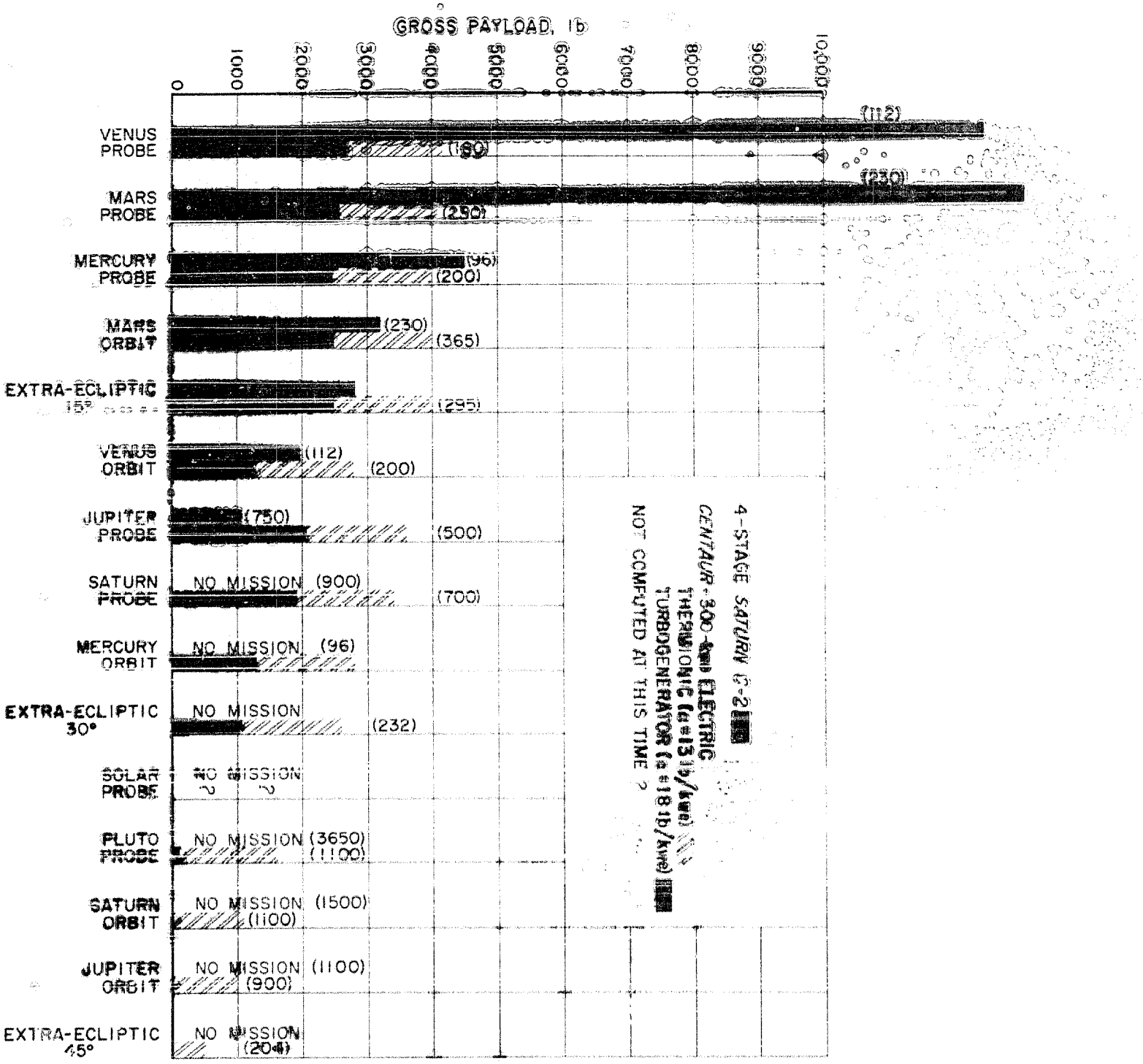

Fig. 2. Performance comparison: *Saturn* chemical vs *Centaur* electric

Table 2. Vehicle capability for the 15 missions

Vehicle type	Number of missions			
	Not feasible	Indeterminate	Uncomputed	Feasible
Four-stage Saturn C-2	8	1	0	6
Centaur, 300 kwe @ 18 lb/kwe (turbogenerator)	4	3	1	7
Centaur, 300 kwe @ 13 lb/kwe (thermionic)	1	2	1	11

The *Centaur* electric, with α of 18, cannot perform four of 15 missions; it cannot be evaluated accurately enough today to determine mission feasibility in three others, while at least seven appear feasible. If the specific weight can be reduced to 13 lbs/kwe, only one of the listed missions is definitely not feasible. Two are indeterminate, and 11 can probably be performed with reasonably good payloads. Thus, it appears that there is strong inducement, from the performance viewpoint, to accelerate work on thermionic reactor conversion systems as the most likely means of achieving the low specific weight needed for planetary and interplanetary missions.

II. SPACECRAFT-SYSTEM COMPARISON OF THERMIONIC AND TURBOGENERATOR NUCLEAR POWER SOURCES FOR ELECTRIC PROPULSION

Although a performance advantage for a spacecraft subsystem always constitutes a strong incentive for its development and adoption, high performance (low component weight) is seldom enough. Often, compatibility of the subsystem with other system components or the ability to utilize it efficiently in the spacecraft will be of overriding importance and the deciding factors in choosing between alternate subsystems. Turbogenerator conversion systems in the form of *Snap VIII* and *Spur* capitalize on existing, or a projection of existing, technology and should be supported. Nevertheless, in attempting to utilize dynamic power-conversion systems, one immediately notes characteristics that are sufficiently troublesome to make a comparison with other potentially interesting systems appear desirable.

This particular comparison between static (thermionic) and dynamic (turbogenerator) nuclear-power-generation systems for electric propulsion will cover a description of the basic systems, a comparison of system characteristics, and a discussion of the influence of the conversion system on over-all spacecraft operation. The analysis and discussion assume that the nuclear heat source is a fast reactor.

Thermoelectric systems were also considered as a means of static power conversion but have been rejected for now. Although acceptable for secondary power, they

appear to be impractically heavy for electric-propulsion purposes. Typical specific weights are about 300 lb/kwe for a 5-kwe generator, according to Ref. 4. Nothing has been found in the literature today that would indicate an ultimate specific weight of less than 20 lb/kwe for the thermoelectric conversion system alone.

A. Description of the Conversion Systems

The dynamic conversion system employs a two-loop Rankine cycle with a nuclear-reactor heat source, as shown in Fig. 3. The reactor is cooled by the primary coolant, lithium, to minimize system pressure and primary pumping power. The fission energy is transferred by the primary coolant to sodium in a shell and tube heat exchanger. The sodium vapor is fed to a turbine assembly, condensed in a waste heat radiator, and returned to the boiler to complete the secondary loop.

The turbogenerator delivers ac power to a transformer-rectifier which provides the desired output to drive the electric thrust unit. If an ion engine is employed, transformation and rectification will probably be required; however, if a magnetohydrodynamic thrust device is used, the generator may be capable of supplying the required electrical output directly.

The reactor exit temperature of 2300°F will provide superheated vapor at a turbine inlet temperature of 2000°F. These temperatures are estimated to be the maximum allowable values commensurate with material limitations over the next decade. A radiator temperature of approximately 1400°F minimizes the required radiator area per unit power.

For comparison, a schematic of the thermionic reactor conversion system is shown in Fig. 4. In essence, the

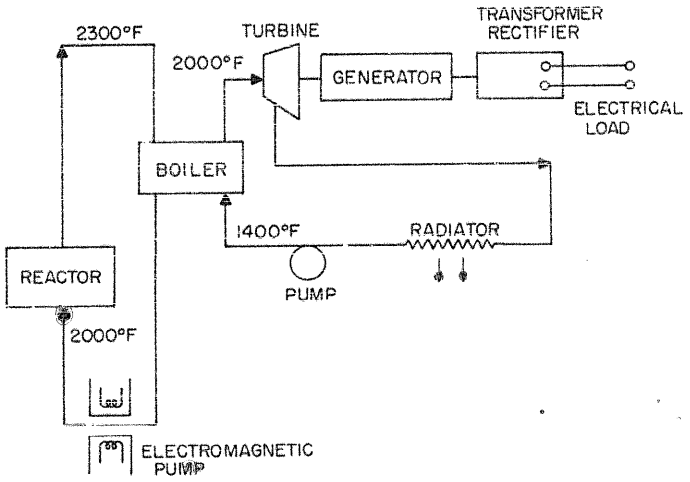

Fig. 3. Nuclear turbogenerator system

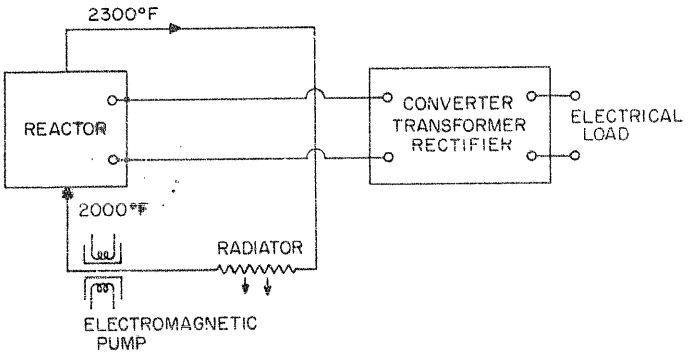

Fig. 4. Nuclear thermionic system

thermionic system utilizes the reactor fuel element as the electron emitter (cathode) and the inner surface of its cooling jacket as the collector (anode) to produce electrical power directly. The collector is electrically (and, to some degree, thermally) insulated from the reactor coolant. The lithium is fed to a waste-heat radiator, where it rejects the excess thermal energy from the reactor and is recycled by use of an electromagnetic pump.

A reactor coolant operating-temperature range equal to that for the turbogenerator reactor (2000 to 2300°F) is assumed. This may not represent an over-all system optimization but, according to Bernatowicz of the Lewis Research Center*, minimizes specific radiator area for an emitter temperature of 3320°F. Since the reactor and radiator operating-temperature range is the same, the radiator area for the thermionic system is much smaller than for the turbogenerator system for the same peak coolant temperature. A specific comparison will be given later.

Series connection of the thermionic diodes could produce dc voltages of 1000 to 10,000 v, depending on the number of separate diodes in the reactor. Practical limitations due to arcing across the core and local discharges in the cesium will probably restrict the output to 100 to 1000 v. It appears that an optimum voltage of approximately 100 v may exist if an inverter-transformer-rectifier is required. Early calculations, based on present technology, indicate that this voltage is optimum from the standpoint of static switching. Since switching represents the major power loss in the converter, this will probably dictate input characteristics.

B. Comparison of System Characteristics

Having summarized the over-all system characteristics, we now consider the inherent advantages and disadvan-

*Private communication.

tages of each of these systems. Both systems have the problem of mass transfer of wall material via the liquid metal coolant. Again, the thermionic system has the advantage of only a single-phase coolant loop, whereas the turbogenerator system utilizes two liquid metal loops—one single-phase, the other, two-phase.

The dynamic system must employ a condensing radiator under free-fall conditions, and although design criteria for such a system should be available in the latter part of this decade, radiator designs may have to be compromised because of this problem. The static system utilizes a single-phase coolant and, thus, circumvents this problem.

If one employs an electromagnetic pump, as shown in Fig. 4, the thermionic system requires no moving parts other than the reactor coolant. It has been shown that the angular momentum associated with this coolant is approximately 1/1000 that due to rotating machinery in a turbogenerator system. The dynamic system, by its very nature, produces high angular momentum as a result of mechanical motion in the turbine and pumps. This motion, superimposed on the high operating temperature, may produce severe creep problems in turbine blades and secondary loop pump impellers.

A weight comparison at the 300-kwe power level shows that the major weight difference between the static and dynamic systems arises from primary radiator weights (columns 1 and 2 in Table 3). This is simply due to the higher radiating temperature of the thermionic system. [If an average radiator temperature of 1540°F (2000°R) is utilized for the thermionic radiator, the reactor and radiator weights are as shown in column 3 for a 20% thermal efficiency.] Based on an over-all efficiency of 15% for the thermionic system and radiator emissivity of 0.9, the specific radiator area is approximately 0.28 ft²/kwe. The comparable turbogenerator system (Fig 3), assuming 20.7% thermal efficiency (0.85 turbine efficiency), has a specific radiator area of 0.83 ft²/kwe for a radiator emissivity of 0.9.

Table 3. Weight breakdown for advanced nuclear turbogenerator and thermionic power systems at 300-kwe power level

Component	Specific weight, lb/kwe		
	Turbogenerator system	Thermionic system	
Reactor* (including pump and controls)	1.9	3.5	2.2
Reactor shield	2.0	3.0	3.0
Primary radiator	7.0	1.8	5.2
Transformer/rectifier (including T/R radiator)	0.9	2.5	2.5
Heat exchanger*	1.5	—	—
Turbine/generator*	1.7	—	—
Generator radiator	0.2	—	—
Total	15.2	10.8	12.9

*The weights of those components which were not discussed in the text have been obtained from estimates by others in the field.

In estimating the weights of the primary radiators for both the thermionic and turbogenerator systems, it was assumed that they consisted entirely of prime surface to minimize radiating area and were designed for a 90% probability of no punctures in one year. For the fixed power level and a 90% probability, the smaller thermionic radiator will weigh 6.5 lb/ft², whereas the larger turbogenerator radiator will weigh 8.5 lb/ft²*. This difference arises, because the turbogenerator radiator requires greater meteoroid protection.

In addition, the dynamic system requires a separate fluid loop to cool the generator. A coolant temperature of 700°F and a generator efficiency of 90% are assumed. The specific radiator area in this case is approximately 0.14 ft²/kwe.

The criterion for calculating the weight per unit area for this radiator is a 90% probability of no spalling in one year. This is required, because any solid material within the coolant may cause clogging of generator coolant passages or freezing of the generator bearings. Based on these assumptions, the weight per unit area is 1.5 lb/ft².

C. *Influence of the Conversion System on the Spacecraft*

We now consider the influence of each of these systems on the over-all spacecraft. Here, the operation of the static and dynamic systems is compared as to its effects on other spacecraft systems, such as attitude control, and electronic component design.

Sketches of a representative *Double Snap VIII* spacecraft configuration and a 300-kwe thermionic configuration, drawn to the same scale, are shown in Fig. 5. These conceptual designs are intended to orient the reader by illustrating major changes that accompany adoption of a more advanced spacecraft rather than to compare 300-kwe turbogenerator- and thermionic-powered spacecraft. Detailed configuration studies will probably alter specific

*The radiator weight of the turbogenerator system may be minimized by increasing the area; for example, the *Spur* radiator will weigh approximately 3.5 lb/kwe at 300 kwe but has an area of 800 ft².

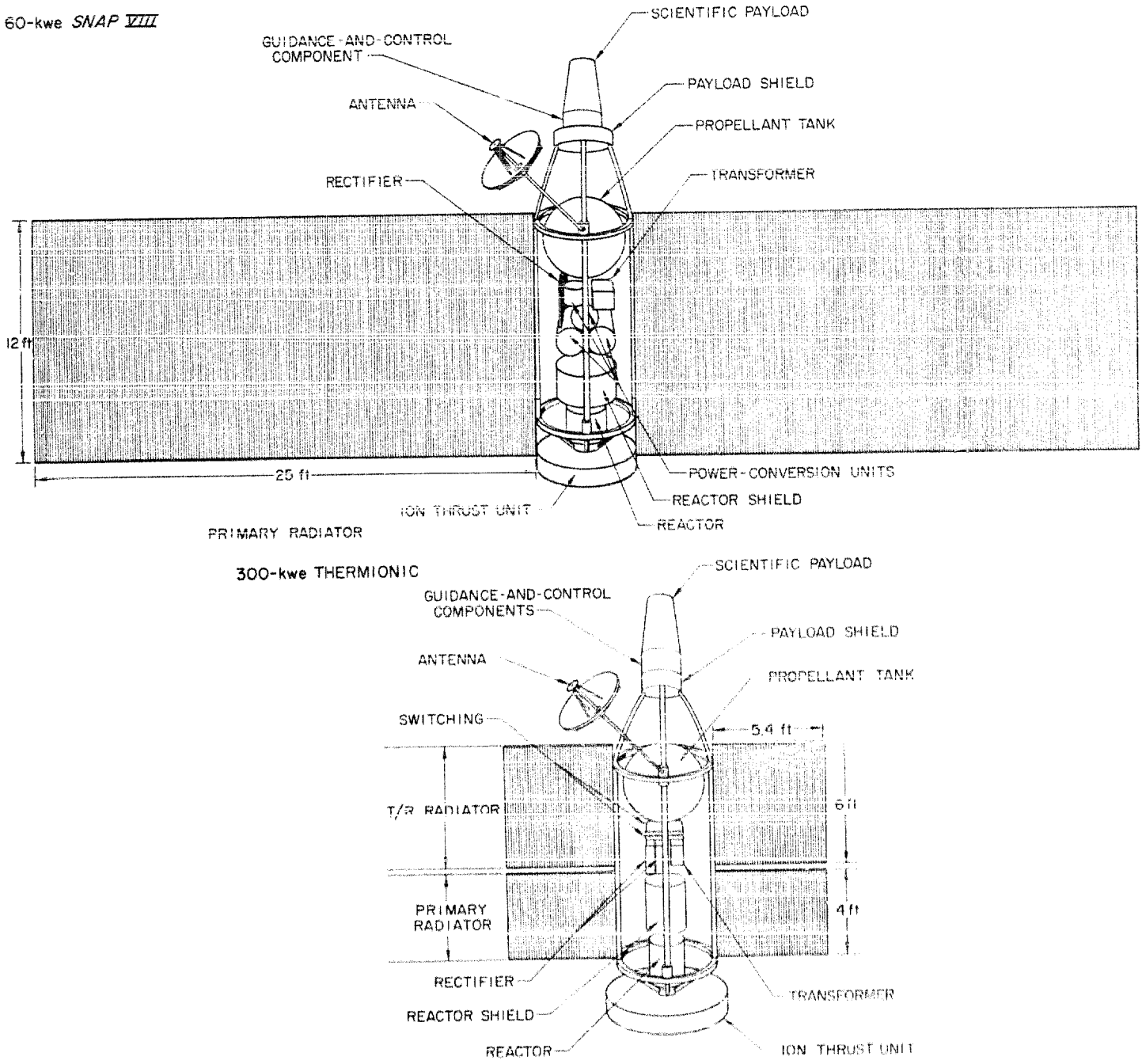

Fig. 5. *Snap VIII* and 300-kwe thermionic spacecraft configurations

design features. Scatter radiation and shielding, especially, must be examined more closely.

Neutron and gamma-ray shielding weights are quite sensitive to component selection and the scientific equipment which is to be carried in the spacecraft. Based on a preliminary analysis, thermionic reactor shield weights are somewhat larger than those of the turbogenerator system for the same allowable dose. This is simply due to the fact that there is a large number of foreign materials in a thermionic-reactor core, which tends to increase fuel-rod size and number, thus increasing the over-all reactor dimensions. A larger diameter shadow shield is required to protect spacecraft components for a given shield thickness.

Assuming a 10-ft separation distance between reactor and sensitive components, 10^{13} nvt fast neutrons in a year,

and a gamma dose of 4×10^5 r/year, the specific weights for the dynamic system shield are 2 to 3 lb/kwe and for the static system shield, 3 to 4 lb/kwe at the 300-kwe level. The comparatively low dosages assumed are taken to provide some degree of conservatism in the weights.

A second important system dependent on the type of nuclear-power-generation system used is the transformer-rectifier (T/R). As discussed previously, if dc to ac inversion is required, the thermionic system suffers a weight penalty as compared to the dynamic system.

Preliminary estimates by T. Koerner of this Laboratory indicate that switching power losses of 2 to 3% and series losses of 1% may be expected, based on present technology. Including transformer and rectifier losses, a total of some 16 kwe may be dissipated in a 300-kwe system. The main problem associated with these losses is the fact that coolant temperatures are limited to approximately 250°F, thus necessitating significant radiator areas to dump this energy.

A comparable system delivering 10,000 v dc to an ion drive utilizing a dynamic system dissipates approximately 5 kwe and requires 50 ft² of radiator for a coolant temperature of 250°F. Using the same radiator criterion as above, the overall T/R specific weight is 0.9 lb/kwe.

A 300-kwe thermionic system, supplying 10,000 v to an ion engine with a reactor dc output of 100 v, requires a 130-ft² radiator. Using a 90% probability of no spalling criterion for meteoroid protection, the over-all T/R specific weight is 2.5 lb/kwe.

At 300 kwe, the thermionic-system primary radiator area is approximately 85 ft², as compared to 250 ft² for the dynamic system. Assuming a flat-plate radiator structure as shown in Fig. 5 (consistent with the weight estimates given above), the natural frequency is 5 cps for the thermionic radiator and 1 cps for the dynamic-system radiator. (For reference, the natural frequency of the *Double Snap VIII* system is about 0.05 cps.) From the viewpoint of attitude control, the higher frequency is preferable; a detailed study may reveal, however, that the lower frequency can be made acceptable to the spacecraft design. If a significant weight penalty accompanies the added complexity of the attitude-control system, further emphasis is given to the development of an advanced reactor system.

The radiator required to cool the T/R also has a natural frequency of the same order as the primary radiator in the thermionic system. This emphasizes the need for high-temperature electronics operation, as the T/R radiator, which dumps 16 kwe, has the same area as the primary radiator, which dumps 1700 kwe.

Since the total radiator area for the thermionic system is only 215 ft², there should be no appreciable packaging problem in spacecraft integration (see Fig. 5). In fact, the total area is only slightly larger than the solar-panel area under consideration for *Mariner B*. Note that the smaller volume associated with the entire thermionic system could be a major advantage for relatively small *(Centaur)* spacecraft, because volume for spacecraft components within the shroud is at a premium.

The turbogenerator system has three separate radiators, as discussed above, with areas of 250, 40, and 50 ft² for the primary generator and T/R radiators, respectively. One possible problem is that of relative orientation. Should the lower-temperature radiators "see" the primary one, their required radiating areas could increase significantly.

A second effect of the smaller radiator area is a combined decreased meteorite-impact probability and smaller torque arm upon impact. Both of these effects will reduce attitude-control requirements directly. If one utilizes a gimballed electric thrust unit for control (which appears to be one of the better methods), the thermionic system minimizes thrust losses.

The problem of damping startup torque due to rotating machinery may provide another significant problem to the attitude-control system. In addition, fluctuations in machinery operation will produce a constant demand on the system. If any, or all, of these problems produce unreasonable demands, the thermionic system may show a substantial net payload advantage over the dynamic system.

Another desirable aspect of the thermionic system is the prospect of operation at partial power, should it be desired. As the fuel temperature dictates the core power output, it should be possible to vary reactor power by a factor of 3 to 8, depending on whether the dominant mode of heat loss from cathode to anode is by conduction or radiation.

The prospect of partial power operation at maximum lifetime efficiency is not so favorable for the dynamic system. Variations in turbine throttle opening produce reactor transients which may be difficult to damp with a fast reactor system.

On–off operation offers problems in storing the working fluid in either system, especially during attempts to reinitiate operation after some dormant period of time. This problem, however, is more complex for the turbogenerator system because of its secondary loop; it will be necessary to maintain the proper operational relationships

while starting up rotating machinery in a loop which contains a solid working fluid that must be melted and vaporized in a free-fall field.

It should be pointed out that both systems may utilize a fast reactor, so that rapid changes in demand might produce undesirable transients. However, in general, the thermionic system offers less complexity to changes in power requirements.

Probably the most important advantage of the thermionic system is its high reliability due to its relative simplicity. Should one cell short out in the reactor, voltage loss is only one volt out of the total series value; with an open circuit, the loss would be about 3½ v. A system predesigned for a certain cell-loss rate should be capable of providing a continuous, steady power output for periods of years, if required.

Although reliability of any piece of equipment is difficult to evaluate until units have been constructed and tested, past experience has shown that properly designed static systems are more reliable than those which contain valves, pumps, and other moving machinery. Because low-thrust propulsion units must operate for unusually long periods of time, this reliability factor will be extremely important in selecting system components.

In summary, the nuclear thermionic system possesses many advantages over the turbogenerator system, both from a subsystem and over-all spacecraft point of view.

III. THERMIONIC ENERGY CONVERSION

Previous Sections of this Report have shown that there is a strong incentive from (1) the spacecraft performance viewpoint and (2) the spacecraft system-integration viewpoint to utilize static and, in particular, thermionic power-conversion systems for future electric propulsion systems. At this point, one might logically ask: What is the present state of the art, and when can we expect such conversion units actually to materialize as flyable hardware?

Considering the fact that the conversion of heat to electrical energy by means of thermionic emission has been seriously considered only in the last four years, the accomplishments to-date are remarkable. The promise of a lightweight, static conversion system has attracted the interest of most of the major electrical and nuclear companies in this country, to the extent that many of them are funding their own programs but at the same time attempting to obtain funding from governmental sources. Several government agencies are funding research and development programs (Ref. 5), including the Army, the Navy, the Air Force, the NASA (on a limited scale to-date), and principally, the AEC, with major efforts at the Los Alamos Scientific Laboratory and Atomics International. The total national annual expenditure for these programs is approximately $30 million.

The technology of thermionic emission is not a new one. Much of the success in thermionics in recent years is attributable to the work, early in this century, of Langmuir as well as that of many others in the fields of basic electronics, atomic physics, and related fields. The

application of thermionic emission as a power-generating device owes a great deal of its prominence to the impetus of this country's accelerated space program. The promise of a lightweight, high-power, long-life, static power source for space missions, as compared to chemical or dynamic systems with their attendant limitations, has elevated this technology from the field of academic interest to one of intense and accelerated research and development, with an eye toward early application.

For long-duration missions, only solar and nuclear prime energy sources are worthy of consideration. For secondary spacecraft power up to approximately 3 to 10 kw, solar thermionic systems offer enough advantages to be competitive with photovoltaic (solar) cell systems, provided the mission is within practical range of the Sun. However, the problems associated with the design and construction of such systems make them impractical for the generation of enough power for propulsion, i.e., 60 kwe to several megawatts.

Regardless of the over-all system power level, the basic diode performance is the ultimate governing criterion for the utility and practicability of the thermionic system. Whether the diodes are to be used with a solar or nuclear heat source also has a great deal to do with their design and performance, because vastly different designs would be required, depending upon the choice of heat source. For the purposes of this discussion, emphasis is placed on their use with a nuclear heat source, although part of the criterion discussed will pertain to the thermionic diode in general, independent of its heat source.

For a better understanding and interpretation of the state of the art of thermionic energy conversion, it is desirable first to discuss the principle of operation of the thermionic diode and to mention how it can be incorporated into a nuclear reactor.

A. Description of Diode Operation

The thermionic diode is based upon the Edison or Richardson effect; i.e., the emission of electrons from hot bodies at a rate which increases rapidly with increasing temperature. By supplying heat to an emitter (cathode) having a low enough work function to give a copious supply of electrons and then receiving these electrons on a lower-temperature collector (anode), whose work function is less than that of the emitter, the excess potential energy of the electrodes is available to perform useful work in passing through an external load. Excess heat must continuously be removed from the collector. During operation, an electronic space charge builds up in the interelectrode space, increasing the potential barrier that the electrons from the emitter must overcome in order to "fall" into the collector.

The two principal methods of overcoming or cancelling this space charge have given rise to the terms vacuum (or close-spaced) diode and cesium (or gas-filled) diode. In the first method, a spacing of 0.0004 in. (10 μ) or less is required between electrodes, with an emitter temperature in the region of 2375°F and a collector temperature of 1000°F, in a vacuum of 10^{-5} mm Hg or lower. Some success has been achieved in laboratory units under well controlled conditions, with efficiencies of 5 to 6% being measured; production units giving 2½% have been made by General Electric. Only about 10% of the national thermionic effort today is in this area.

Much higher efficiencies and improved performance are attainable with devices using the second method for overcoming space charge. This is accomplished by using an easily ionized gas, such as cesium, in the interelectrode space. The cesium ionizes upon contact with the hot emitter (whose work function is higher than the first ionization potential of cesium), providing positive ions for neutralizing the space charge; this permits an increased flow of electrons across the gap. Close-spacing is not required in this instance, thus avoiding one of the major difficulties of the close-spaced vacuum diode. Recent work has shown however, that with low cesium pressure and close-spacing, say 0.010 to ¼ mm, higher power outputs and efficiencies can be achieved. Efficiencies as high as 20% have now been attained experimentally in the laboratory, and efficiencies of 30% are theoretically possible. It is not unreasonable to predict, therefore, that converter efficiencies of 20% will be achieved in practice, with emitter and collector temperatures of 3000°F and 1850°F, respectively. It should be pointed out that, depending upon the emitter and collector materials and temperature, as well as the cesium pressure and spacing, the cesium does more than just neutralize the space charge. It may also coat the emitter to varying degrees, lower its effective work function, and thus provide more electrons; similarly, coating of the collector would lower its work function to that of the cesium.

B. Incorporation of a Diode Into a Reactor

To utilize a thermionic diode in a reactor design, one need only recognize that fission heat is adequate to heat

the emitter to an appropriate temperature and that the collector could be cooled by a reactor coolant flowing external to it. By proper design, a dc output of several hundred (or possibly more) volts could be achieved merely by placing as many diodes in series as are needed to obtain that voltage.

The reactor itself would probably be a fast reactor. As calculations show, a core size of about 9 to 12 in. in diameter by about 12 to 18 in. in length is sufficient to achieve criticality with a power density of the order of 4 mw (thermal) per cubic foot. The over-all reactor envelope would be of the order of 2 ft in diameter by 3 ft long and would be capable of operation in the one-megawatt (electric) range. Including reflector, pump, controls, and radiator, but excluding shielding, a reactor specific weight of about 4 lb/kwe is believed to be achievable.

Illustrative of a typical thermionic reactor design is that of Atomics International, taken from Ref. 6 and shown in Fig. 6. Details of the fuel element for this design are shown in Fig. 7.

There are two philosophies regarding the emitter design. The first assumes that the emitter surface material is some type of uranium fuel. UC-ZrC is one of the most promising because of its excellent emission properties; UC provides the low work function for emission, and ZrC provides the high-temperature strength. One thus has a relatively simple, compact, unclad fuel element; associated with this design, however, are problems of vaporization and fission-product emission.

The other approach is to clad the fuel, which could now be UC, or any other suitable high-temperature uranium compound or mixture. The cladding would be a

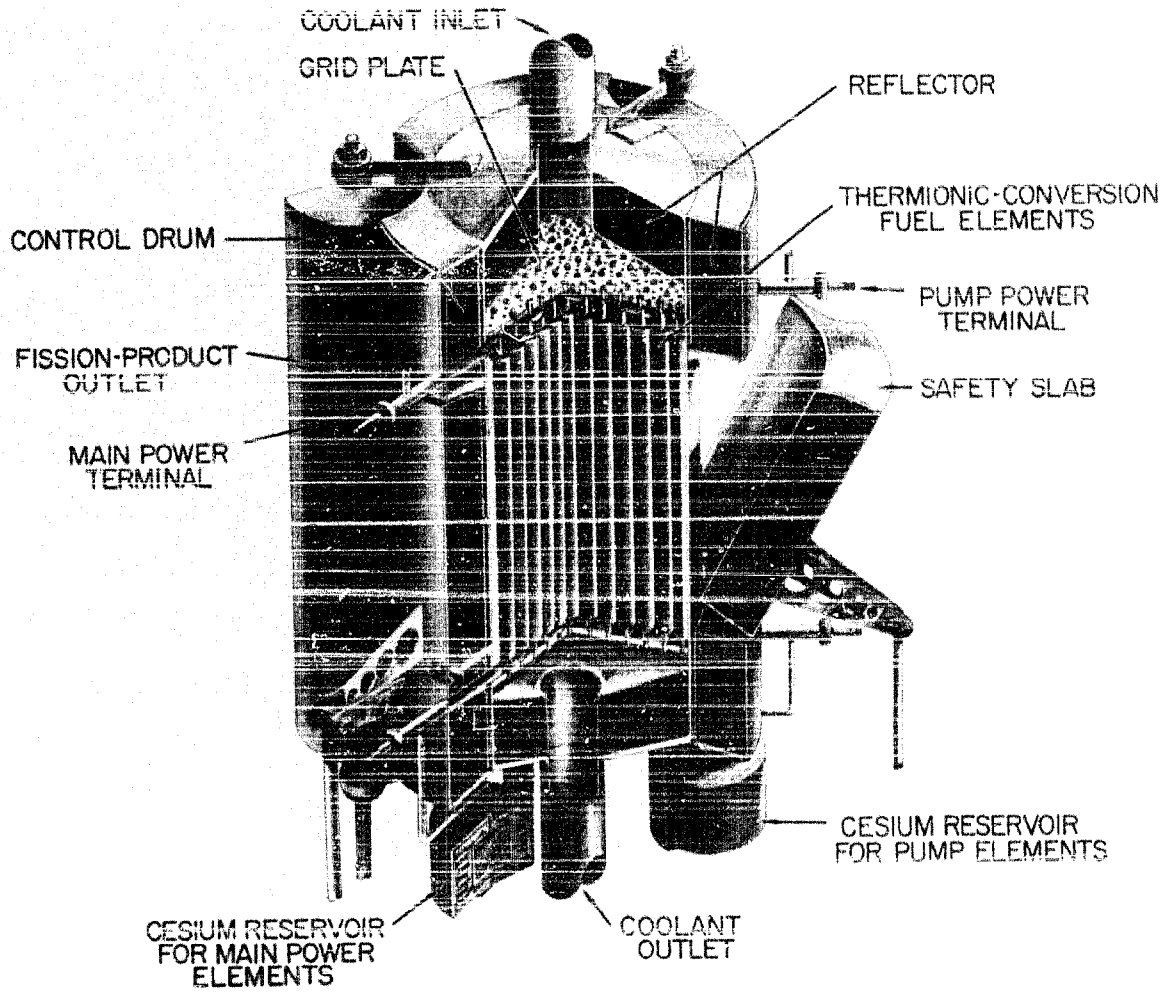

Fig. 6. Thermionic reactor

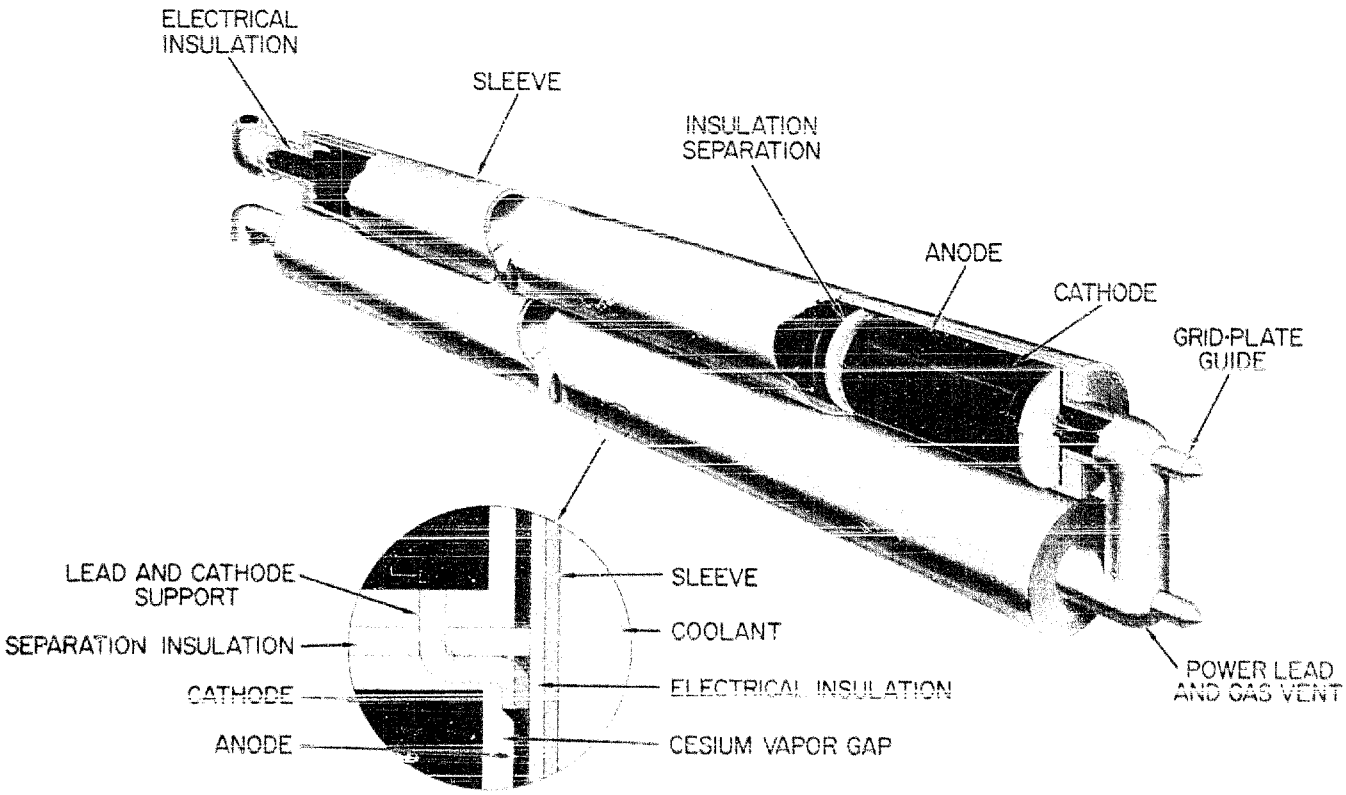

Fig. 7. Thermionic-conversion fuel element

refractory metal with good emission characteristics as well as high-temperature stability and strength. The clad element would probably require a higher maximum fuel-pin temperature than the unclad element to produce the same emitter surface temperature. Care must be taken to insure integral contact between the fuel and its cladding to avoid any increase in ΔT between the fissioning fuel and the emitting surface. The cladding would also increase the non fuel content within the core and would therefore require a slightly larger reactor in order to provide enough fuel for a critical mass.

C. State of the Art

The above discussion is intended to illustrate the type of thinking and nature of the problems currently associated with thermionic conversion work. At present, we are at a point of recognizing such problems, and an effort is being directed toward resolving them. Though much work has yet to be done in the research laboratory (see Ref. 7), these and other materials and engineering problems are well on their way toward solutions.

Table 4, which is representative of opinion in the field, illustrates current experimental data for thermionic components in the laboratory. Included also are performance estimates felt by many to be attainable in a practical operating system. Opinion, as usual, is rather mixed on the optimism of the estimated performance; however, it is believed that the values represent a realistic estimate of what can be achieved on an appropriate time schedule.

Table 4. Current and predicted future thermionic converter characteristics

	Current experimental data	Predicted future performance
Power flux	10 w/cm^2	20 w/cm^2
Emitter temperature	3100°F (unfueled)	3600°F
Collector temperature	1100–1300°F	1650°F*
Cesium temperature	525°F	575°F
Gap	0.030 in.	0.020 in.
Voltage output	1 v	1 v
Current flux	10 amp/cm^2	20 amp/cm^2
Over-all efficiency	15%	20%*

*These values for collector temperature and over all diode efficiency are predicted for optimum converter design. They differ from the 2300°F temperature and 15% efficiency used in the section on the turbogenerator–thermionic comparison, because the latter attempted to optimize the over-all spacecraft system; i.e., a higher collector temperature permits a smaller radiator but results in lower thermionic efficiency.

D. Major Problem Areas and Status

From the point of view of providing electrical power for space propulsion, there are three principal areas of development which must be pursued:

1. Thermionic converter
2. Reactor plus converter
3. Reactor spacecraft system

Because of the necessity of optimizing the entire system, the system approach must be taken in the development and design of both the converter and the thermionic reactor. Despite many problems peculiar to each system component, there are some problem areas which are common to the system as a whole. The two most obvious and important areas are:

1. Materials development
2. Reactor design and operation

Solutions to the materials problems are the first priority in this program, but at the same time, the design and operational problems have an important influence on the ultimate success of the system. Typical of the materials problems requiring solution, all of which are currently under intensive investigation to the extent that limited funding allows, is the development of the following:

1. Electron emitters with an optimum work function, a high ratio of electron emission to atom evaporation, low surface thermal emissivity, and long life
2. Electron collectors with a low work function, good absorption of cesium, low back emission, low thermal emissivity, and long life
3. High-temperature cesium-resistant and radiation-resistant ceramic to metal seals
4. Nuclear thermionic fuel with good emission properties, low diffusion coefficient for uranium and fission products, and stability under irradiation

Typical of the major design and operational problems are:

1. Design complexity associated with incorporation of a thermionic converter into a reactor core
2. Design and development of a thermionic fuel element
3. Power density limitations due to emission-limited emitter
4. Nonuniform power and temperature distribution over core volume, leading to variations in power output and efficiency
5. Control of an essentially fixed temperature and fixed output power reactor of optimum design at these points and at partial loads
6. Lifetime and reliability of operation as a function of lifetime and reliability of individual components

The last of these is extremely significant and will not only influence the design and operation of the system but is a large factor in the materials development program as well.

In summary, it is becoming increasingly evident that work on thermionic reactor systems is in the development stage, and applications must now be seriously considered in both the planning and actual work. Although additional research studies are needed on thermionic materials, converter technology, and design, the concept of thermionic power conversion has been established, operating devices of relatively high efficiency have been built, and some preliminary design studies have been made, indicating possible conceptual thermionic reactor designs, system weights, power outputs, efficiencies, etc. There seem to be no insurmountable problems in the development of the thermionic reactor.

There is optimism on the part of many engaged in the development of this concept that a flight-type thermionic reactor system can be developed and available for use by 1968, provided that proper emphasis and adequate funding are made available in the near future—plus a definitive mission requirement toward which to work and upon which to base specific system design.

IV. SUMMARY AND CONCLUSIONS

Although electric propulsion systems will undoubtediy be used for other purposes, they will probably find their greatest area of application as primary propulsion for planetary and interplanetary missions. In the case of vehicle attitude control, vehicle orbital correction, and Earth orbital transfers (e.g., to a 24-hr orbit), or for lunar missions, they must compete with chemical, direct nuclear, or possibly other propulsion schemes, and a choice may be difficult to make. In the very high-energy interplanetary missions, however, the increasing importance of the electric systems' high specific impulse quickly becomes apparent.

The studies described in this Report show that 50% of the unmanned planetary and interplanetary missions of major scientific interest examined cannot be performed by chemical systems based on *Saturn*, and approximately 40% cannot be performed by *Nova* or by nuclear heat-exchanger systems of the Rover type. Possible modifications, such as placing a fifth chemical stage on a *Saturn C-2*, do not change this situation basically. On the other hand, all of these missions can be performed with an electric-powered spacecraft which uses a *Saturn* booster; most can be performed by systems utilizing an electric-powered spacecraft with a *Centaur* or *Advanced Centaur* booster.

If a vehicle with a payload capability of 15,000 lb in 300-nm Earth orbit, such as the *Advanced Centaur*, were to become available, it could prove to be NASA's most suitable vehicle, from the technical viewpoint, for the planetary–interplanetary missions in the period 1970 to about 1980. Such a vehicle would appear to avoid the standard *Centaur's* marginal payloads for the very difficult missions and the excessively large payloads of the more costly *Saturn* electric.

Of the possible electric propulsion systems, those employing thermionic conversion of reactor power appear at this time to offer strong advantages in both performance and system integration. The lower power-plant weight expected, compared to turbogenerator plants, will be of considerable value for the higher-energy missions. System reliability will probably be greater, because the propulsion subsystem will have no moving parts. Attitude-control problems will be simplified by the freedom from the large gyroscopic moments associated with turbogenerators and the smaller size of radiators associated with higher radiator temperatures. Both restarting and operation at reduced power levels for long-time communication seem much simpler with thermionic conversion.

For an electric propulsion system useful for planetary and interplanetary missions, a power level of about 300 to 1500 kwe delivered to the drive unit seems desirable for the next decade. The *Snap VIII* or *Double Snap VIII* power plants, because of their lower power outputs and high weight per kilowatt, offer little likelihood of performing planetary or interplanetary missions that cannot be accomplished by chemical systems during the same period of time. Because of the large radiators and the rotating machinery which they employ, use of these power plants will require that designers of the spacecraft systems find solutions to serious engineering problems, which may be circumvented with future electric systems.

In spacecraft systems using electric propulsion, system-integration problems appear even more complex than in systems using chemical propulsion. The high temperatures required for efficient heat utilization and rejection in the power source may make it advisable to use power transformers, rectifiers, and inverters or alternators that can operate in such a high-temperature environment; this may prove to be equally true for the electronic components. The presence of nuclear radiation provides another example of strong system interactions. Problems with large, rigid, lightweight structures and attitude-control problems with "floppy" radiators and rotating machinery may also arise. High electrical currents in the power system, if not controlled carefully, can create magnetic fields, which could invalidate magnetometer experiments or induce large and destructive currents in adjacent electronic components. Subsystems, including the propulsion subsystem, must from the start be guided by and closely linked with the over-all system design.

It is believed that feasibility of a nuclear thermionic electric power subsystem can be established within one to two years if a serious attempt to do so is made now. The major problems are those of materials development, development of a thermionic fuel element, and establishment of operation lifetime. A large majority of the problems have to do with materials work which must be supported adequately if the potential of the thermionic system is to be fully realized.

If feasibility is established by 1962–63, it is believed that a spacecraft system utilizing nuclear thermionic electric propulsion power could be flown on a high-energy mission (such as an extra-ecliptic probe at 35 deg.

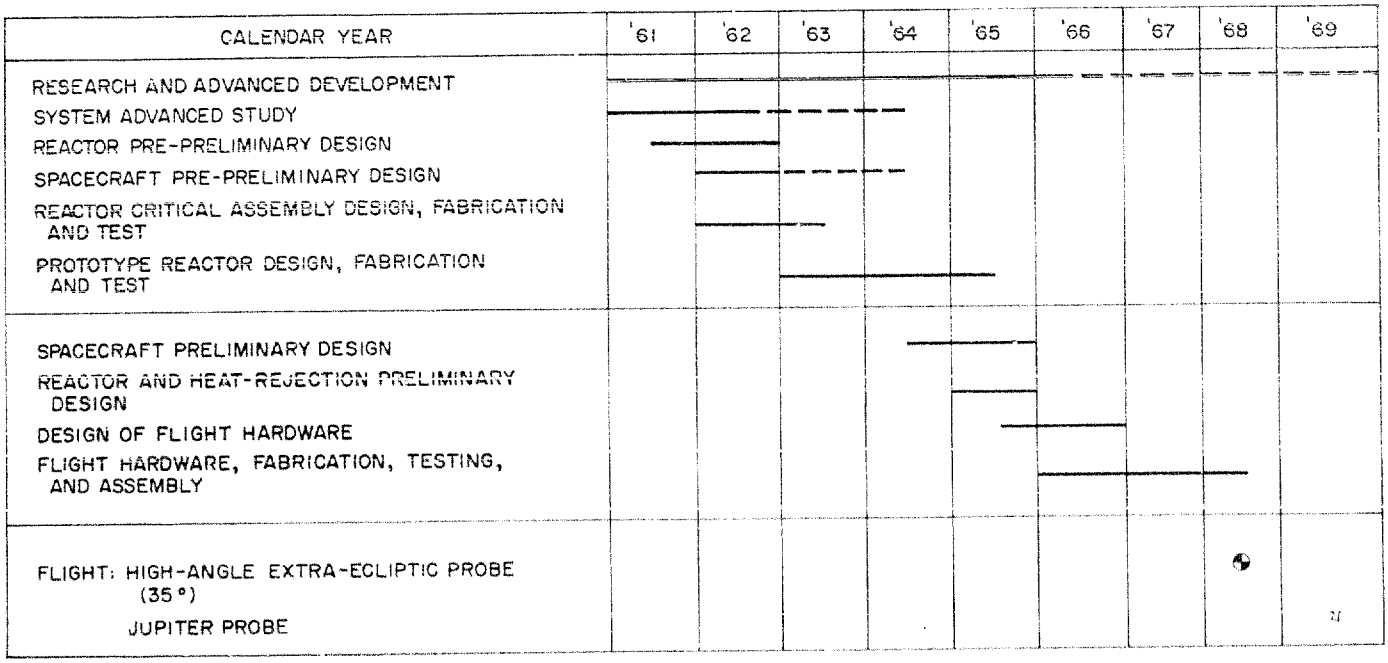

Fig. 8. NuSTEP schedule

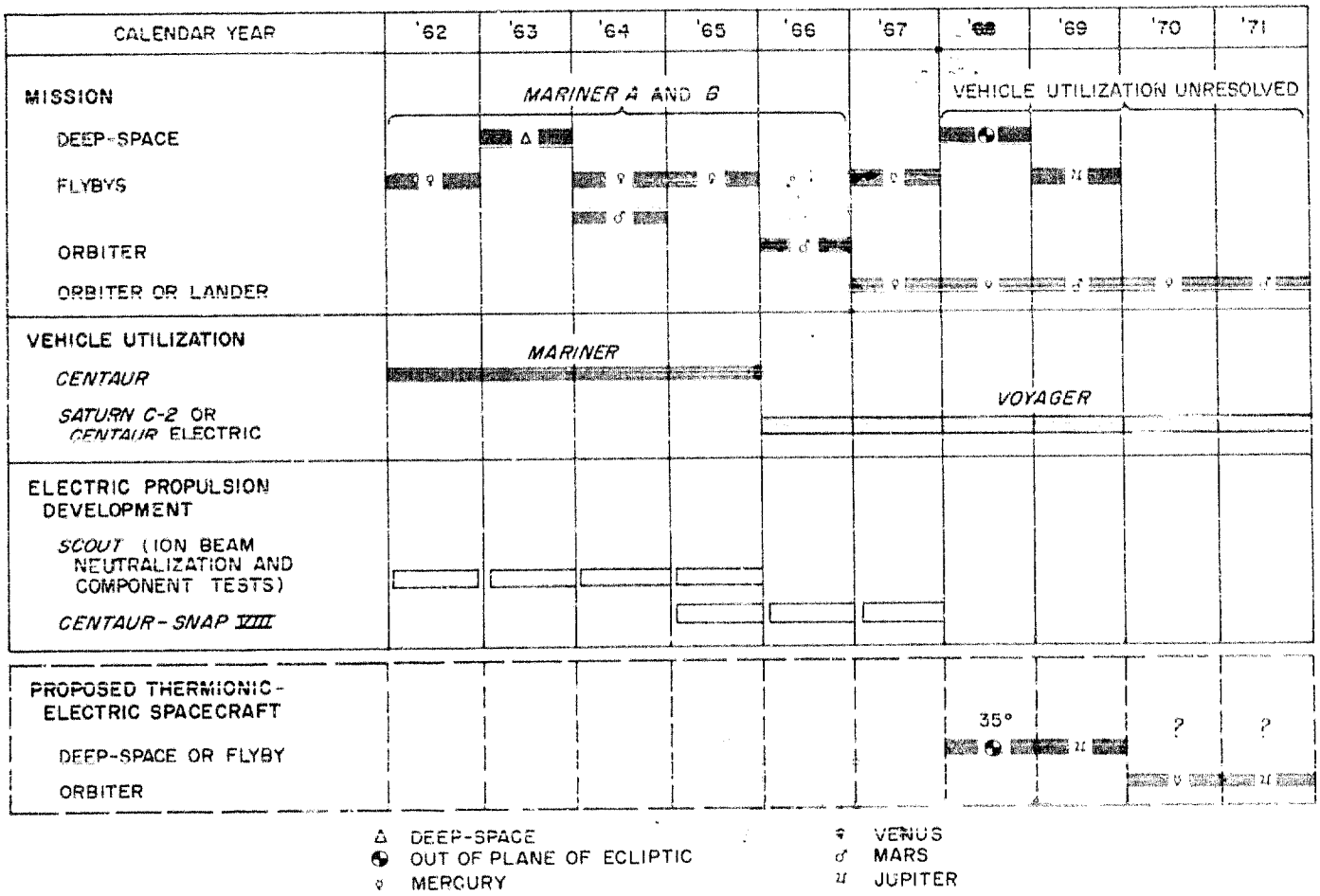

Fig. 9. Planetary–interplanetary program mission plan

an extremely important scientific mission) by 1968–69, when such a mission could not be carried out by any other means. This mission, with its simple operational procedures, is especially desirable, because early missions in a new project should consist principally of a checkout of the spacecraft system. Soon afterward, a Mercury orbiter and a Jupiter orbiter, missions also uniquely suitable to electric propulsion capability, will be desirable objectives and could be scheduled.

Figure 8 indicates a preliminary schedule of major milestones in the development of a spacecraft based on nuclear systems for thermionic electric propulsion (NuSTEP). Although tentative, it is valuable for planning purposes in that it indicates when each phase of the work must be started. Figure 9 shows NASA's Long-Range Planetary–Interplanetary Program for reference purposes; development flights in NASA's electric propulsion program have been incorporated as a matter of interest.

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